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EFFECTS OF BONDING TEMPERATURES ON FATIGUE CRACK GROWTH

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Structrual Integrity Branch
Structures and Dynamics Division

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Changes in the crack growth behavior of 7075-T651 aluminum specimens which had been exposed to temperatures between 150° and 355°F (66° and 179°C) were evaluated. Specimens were fatigue tested at room temperature under flight-by-flight loading conditions. Results from these tests were then compared with data from the baseline (as received) material. Exposure to 250°F (121°C) or more produced a definite increase in specimen life, apparently due to a decrease in crack growth rates.

FOREWORD

This report describes an in-house effort conducted under Project 2041, "Structures and Dynamics," Task 240101, "Structural Integrity for Military Aerospace Vehicles," Work Unit 24010109, "Life Analysis and Design Methods for Aerospace Structure." The report is an expanded version of AFWAL-TM-82-191-FIBE, which was published in June 1982.

The work was performed for the Structural Integrity Branch, Structures and Dynamics Division, Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories (AFWAL/FIBE), Wright-Patterson Air Force Base, Ohio. The research was conducted under the direction of Lieutenant R. L. Wilkinson and Mr. J. M. Potter from May 1981 through August 1982. Dr. J. M. Papazian, Grumman Aerospace Corporation, provided technical assistance in the area of microstructural effects.

The authors wish to recognize Mr. Harold Stalnaker for his advice and assistance in conducting fatigue tests, Mr. Richard Kleismit for heat-treating specimens, Mr. Jack Smith for conducting tensile tests, and Mr. Larry Bates for preparing specimens and assisting in all of the above areas.

The completed report was submitted in February 1983.

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SECTION I

INTRODUCTION

One of the devices being developed for individual aircraft tracking is called the "crack growth gage." The crack growth gage is a small cracked metal coupon which, in operation, is attached to a load-bearing aircraft structural member. The theory behind the gage is that it will experience the same loading environment as the critical structural element and thus, any monitored growth in the gage will be proportional to that in the critical element (References 1, 2). In operation, crack growth at different critical structural details would be related to crack growth in the gage by the development of a "transfer function" for each detail. The crack growth gage is projected to be the primary structural monitoring device. Therefore, it must be extremely reliable; inaccuracies and inconsistencies can lead to excessive, costly maintenance or worse yet, to a "safe" indication on an airframe which may quickly be growing dangerous structural cracks. Unfortunately, development tests for the crack growth gage have proven inconclusive because of a large amount of scatter in crack growth data (References 3, 4, 5). As part of the Holloway tests (Reference 5), several factors were investigated and determined not to be responsible for the crack growth variation. Factors checked were stress in carrier specimen, stress in crack growth gage, load transfer to the gage over the duration of the test, and bending in the gage. A factor which was not considered during these tests was the temperature of the adhesive cure cycle and its possible effect on the crack growth gage material. The adhesive cure cycle temperatures typically exceeded 325°F (163°C), and could have had a considerable metallurgical effect on the 7075-T6 and 7075-T651 materials used in References 1-3. These materials are artificially aged at only 240° to 260°F (116° to 127°C). Since no crack growth data could be found for 7075-T6xx materials which had been subjected to short term heat cycles, the authors decided to generate these data to determine if the adhesive cure cycles could be the source of crack growth gage variations.

The purpose of this program was to evaluate changes in the crack growth behavior of 7075-T651 aluminum specimens which had been exposed to elevated temperatures. Center-cracked panels were subjected to temperatures with maximums between 150° and 355°F (66° and 179°C), cooled, and fatigue tested under variable amplitude loading. Crack lengths were visually monitored and periodically recorded. Results from these tests were then compared with data from the baseline (as received) material.

SECTION II

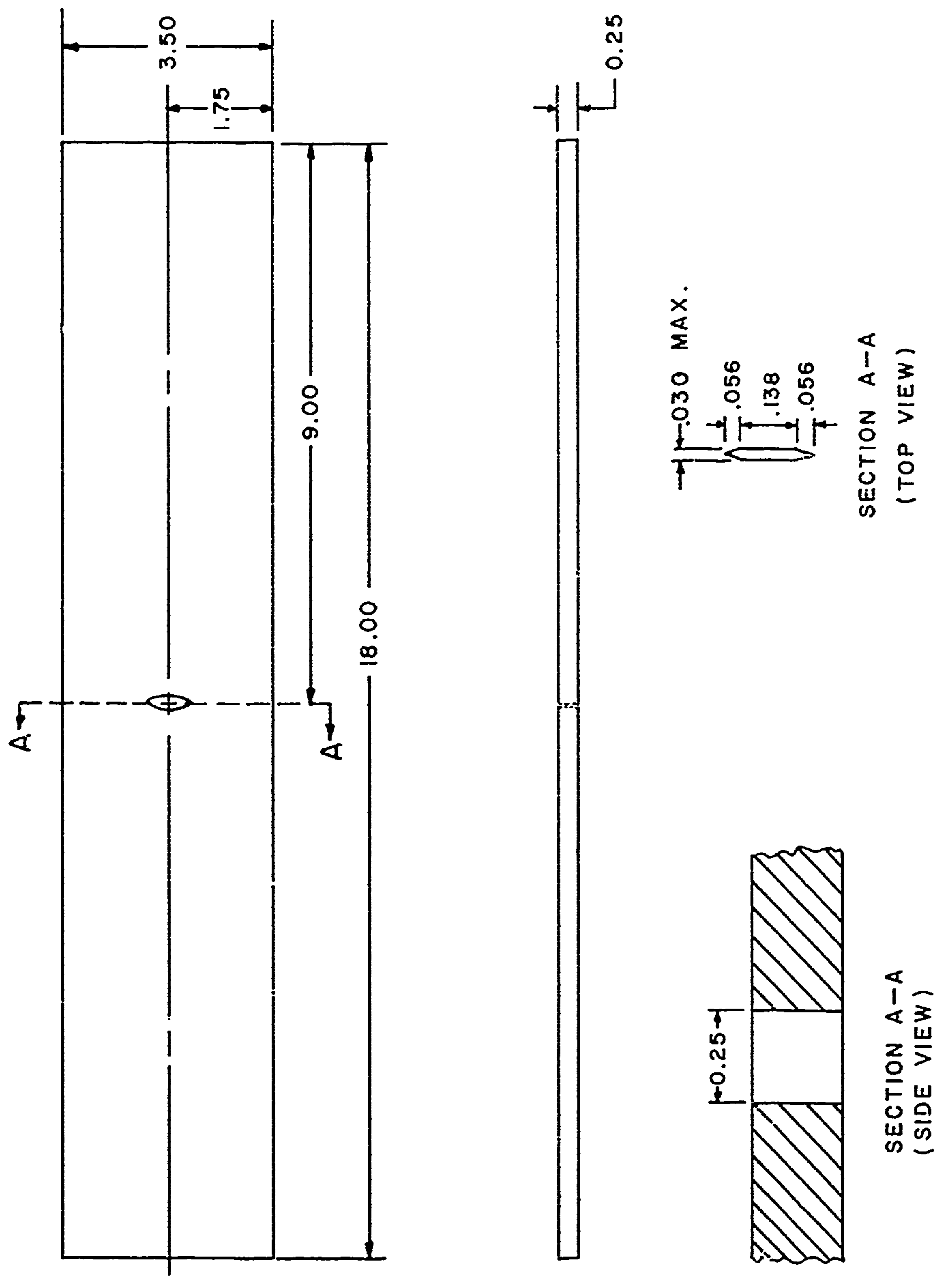
EXPERIMENTAL PROCEDURE

1. SPECIMEN PREPARATION

The material used for this program was taken from a 0.25 inch (6.4 mm) thick plate of 7075-T651 aluminum. This alloy was selected because it is a candidate material for use in crack growth gages and because its fatigue behavior has been widely studied. The 0.25 inch thickness was chosen for convenience since the magnitude of any effects present in this configuration should be equivalent or greater in the thinner (0.04-0.06 in., 1.0-1.5 mm) crack gage sections. Fatigue test specimens were center-cracked panels as shown in Figure 1, manufactured in accordance with ASTM STD E-647 and oriented such that crack propagation was in the LT direction. Slots were introduced by electro-discharge machining (EDM). Specimens were not precracked prior to the start of fatigue testing.

After machining, the specimens were exposed to short-term heating cycles chosen to represent various bonding procedures (Table 1). These cycles were based on documented practice (Reference 3) and standard laboratory bonding procedures for American Cyanamid's FM-73 adhesive. Other heat cycles were evaluated (Table 2), but are not discussed in detail because they did not produce significant changes in specimen behavior.

Specimens were heated in a laboratory convection oven. A technician monitored the oven air temperature and kept it within 5 degrees (3°C) of the specified values. Temperatures listed in Tables 1 and 2 were obtained from thermocouples which were placed on the specimen surface, covered, and held in place by weights. All specimens were heated and cooled at rates between 5 and 7°F/minute (3 and 4°C/minute). The maximum time any specimen took to reach the control temperature was 40 minutes.



NOTE: ALL DIMENSIONS GIVEN IN INCHES.

Figure 1. Specimen Geometry

Table 1: Heat Cycles Chosen To Represent Typical Bonding Procedures

Temperature °F (°C)	Baseline*	250 ± 2 (121 ± 1)		285 ± 2 (141 ± 1)		320 ± 3 (160 ± 2)		355 ± 3 (179 ± 2)	
Time At Temperature (Min ± 1)	As Received	120	60	120	60	120	60	120	60
Specimen Numbers	B-5 B-6 B-7 B-8	4A-1 4A-2	4B-1 4B-2	3A-1 3A-2 3A-3	3B-1 3B-2	2A-1 2A-2	2B-1 2B-2	1A-1 1A-2	1B-1 1B-2

*Room Temperature: 75 °F (24 °C)

Table 2: "Less Severe" Heat Cycles Which Were Evaluated

Temperature °F (°C)	250 ± 2 (121 ± 1)	235 ± 1 (113 ± 1)	200 ± 1 (93 ± 1)	150 ± 1 (66 ± 1)
Time At Temperature (Min ±1)	10	60	60	60
Specimen Numbers	4C-1 4C-2	6B-1 6B-2	8B-1	9B-1 9B-2

2. FATIGUE TESTING

Each specimen was individually fatigue tested in one of three servo-controlled axial loading frames. All testing was done in laboratory air at 75°F (24°C) and 50% humidity. The load history consisted of random flight-by-flight loads, with each repeat of the history comprising 400 equivalent flight hours. It was derived from the F-16 lower wingskin load history previously used by Noronha, et al. (Reference 6). The maximum stress, based on gross section area, was 29 Ksi (200 MPa) and negative loads were clipped at zero. Loads were applied at an average rate of 2 Hz. For more information on the load history, see Appendix A.

Crack lengths were visually monitored and recorded every 400 flight hours. Technicians used low power stereo microscopes and transparent scales to obtain crack length measurements with an accuracy of $\pm$ 0.002 inches. All values for crack length listed in this report refer to the total crack length (2a) measured from tip to tip. Data were not smoothed or filtered.

SECTION III

RESULTS

1. CRACK GROWTH LIFE

At the completion of each test, a crack growth life was calculated based on the time required for a 0.3 inch (7.62 mm) crack to grow to failure. The starting crack length of 0.3 inches was selected arbitrarily to remove the effects of crack initiation. The corresponding number of flight hours at that point was estimated by linearly interpolating between available readings. No specimen contained an initial notch longer than 0.258 inches (6.55 mm). All observed crack growth was symmetric about the notch.

Specimens exposed to elevated temperatures consistently demonstrated longer lives than "as received" specimens. While the average crack growth life for baseline specimens was 6350 flight hours, specimens exposed at 355°F (179°C) lasted an average of 9300 flight hours -- an increase of nearly 50 percent (Figure 2). Total fatigue life was also evaluated, but initiation times showed no significant changes as a function of thermal exposure. The average increase in total fatigue life after the 355°F exposure was approximately 35 percent.

2. IMPORTANCE OF EXPOSURE TIME

Although the effects of one and two hour exposures appear to be slightly different (Figure 2), the data collected do not indicate that this difference is significant. The remainder of this report will focus on exposure temperature only, with each data point representing the average of four specimens (2 one-hour exposures and 2 two-hour exposures). Complete data lists for all specimens are included in Appendix B.

3. CRACK GROWTH RATE

Figures 3 through 6 show the average crack growth rates observed for the four specimens of each temperature group compared with data obtained for the baseline specimens. Growth rate data were plotted in terms of ΔK_{RMS} using the relationship

$$\Delta K_{RMS} = \Delta\sigma_{RMS} \sqrt{\pi a}\ \beta \qquad (1)$$

where

$$\Delta\sigma_{RMS} = \left(\frac{\sum_{i=1}^{n} (\sigma_{max_i} - \sigma_{min_i})^2}{n} \right)^{1/2}$$

$$\beta = 1 + 0.256\left(\frac{a}{w}\right) - 1.354\left(\frac{a}{w}\right)^2 + 12.19\left(\frac{a}{w}\right)^3$$

n = Total Number of Cycles in Spectrum

a = Half Crack Length (inches)

w = Specimen Width (inches)

Straight-line curve fits were added using least squares linear regression. Data for the non-linear portion of the da/dF curve (1.3×10^{-5} and below) were not included in these plots. For the 320°F and 355 °F exposures (Figures 5 & 6), crack growth rate was a major factor in the longer specimen lives. The slope of the da/dF curve for these two conditions decreased 14% and 19%, respectively, from the baseline da/dF slope. Lower temperatures however, did not appear to significantly affect crack growth rate. Exposure at 285°F caused no noticeable change in da/dF slope, while the 250°F exposure actually increased the slope slightly. Increases in specimen life corresponding to these exposures must have been related to some other factor (such as toughness).

4. MATERIAL PROPERTIES vs LIFE

Hardness, yield strength, and ultimate strength tests were conducted in an attempt to relate changes in crack growth life to some tangible

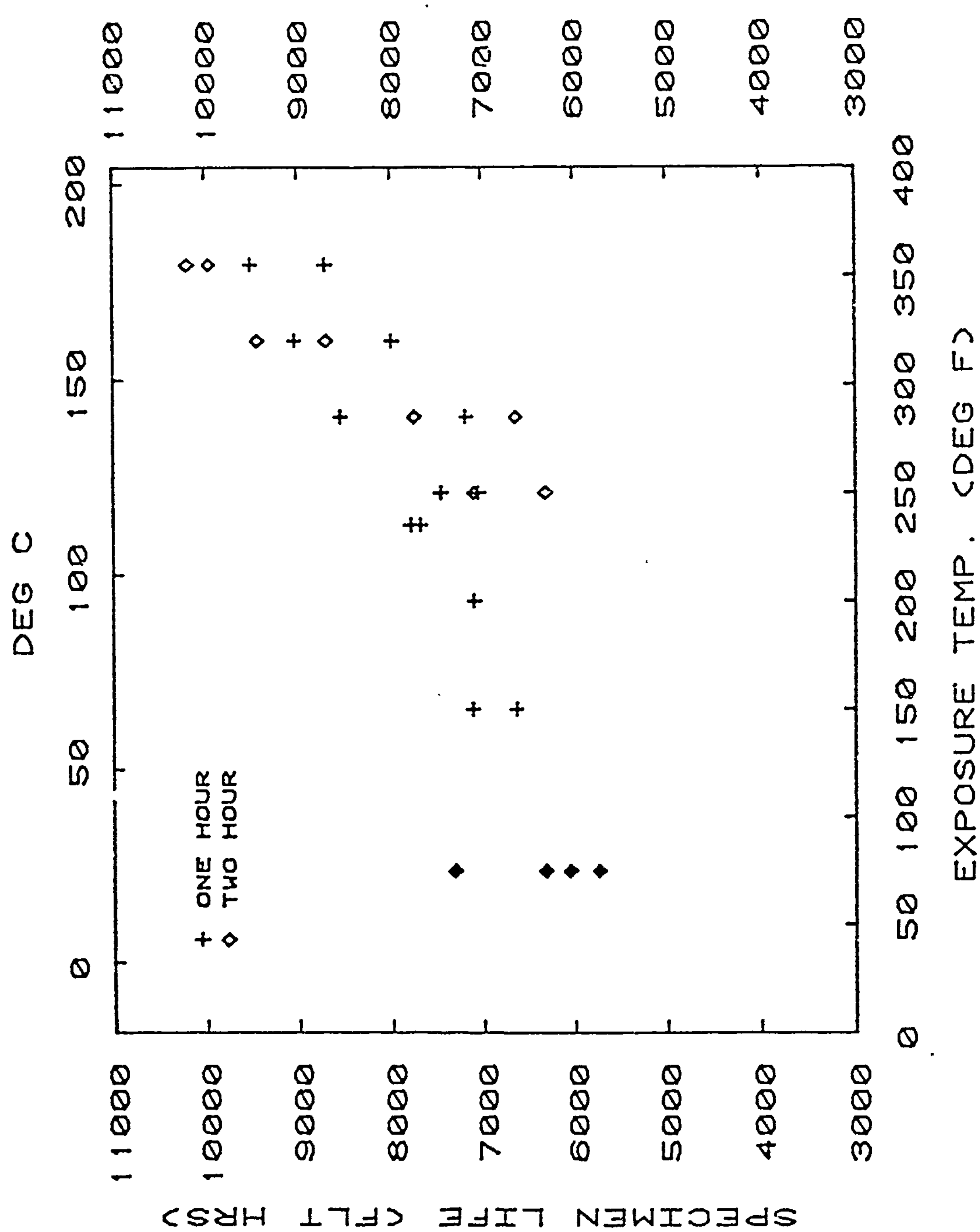


Figure 2. Specimen Crack Growth Lives

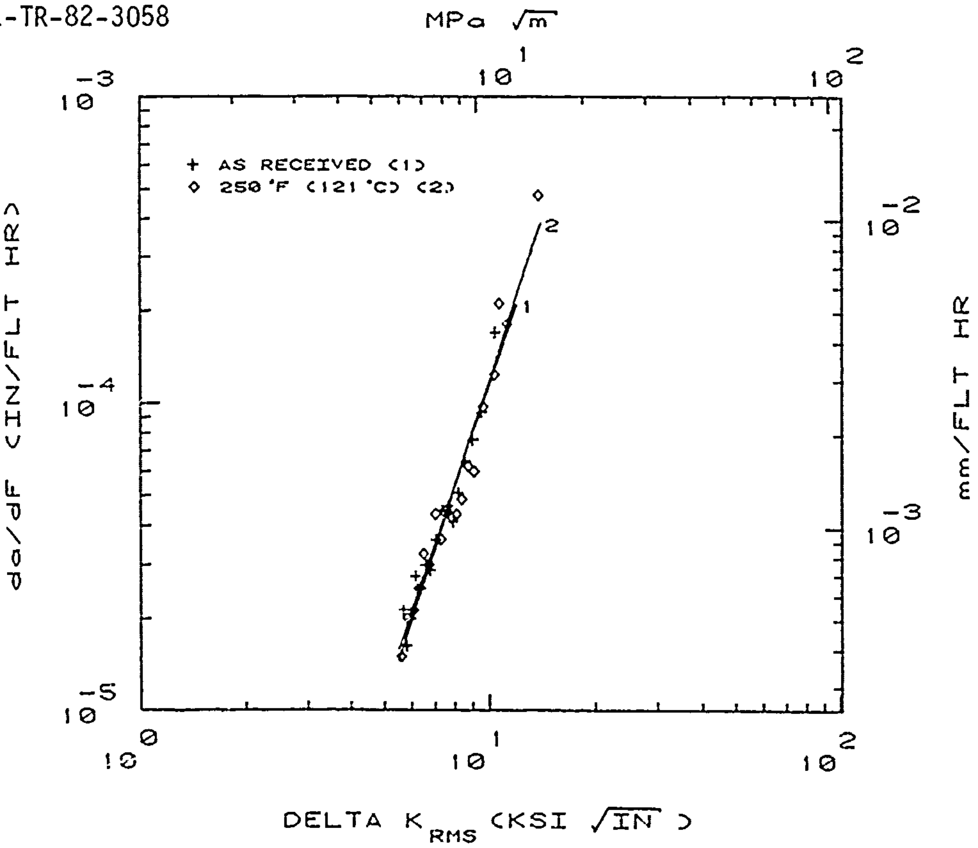


Figure 3. Crack Growth Rates After Exposure at 250°F (121°C)

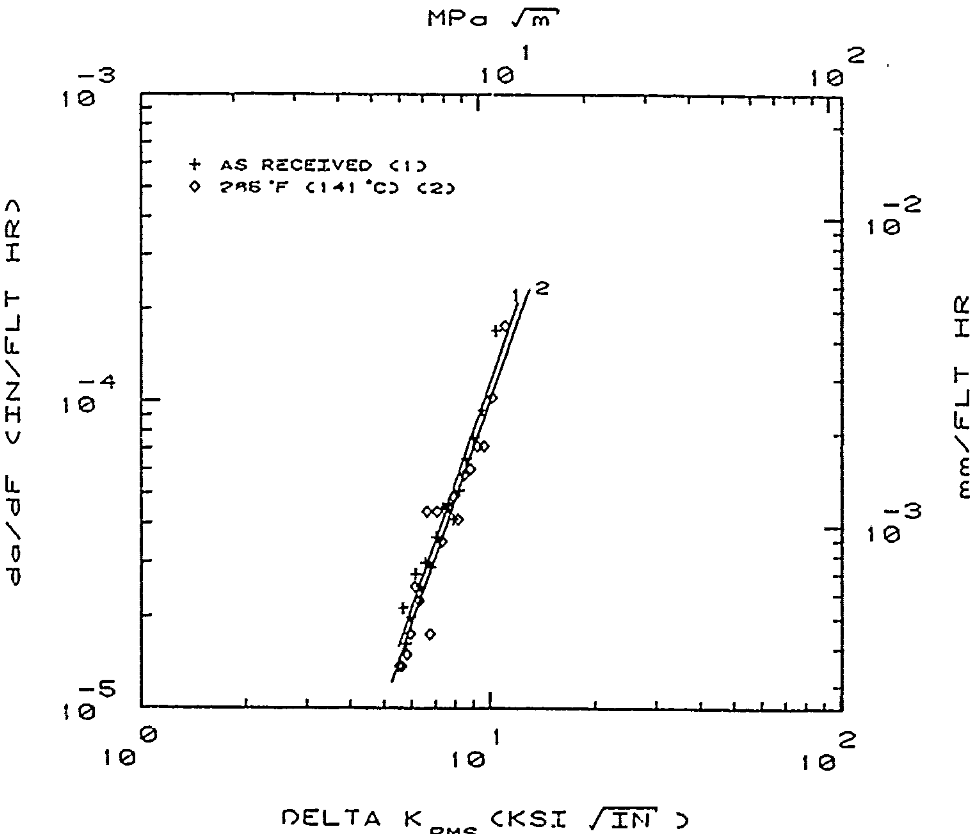


Figure 4. Crack Growth Rates After Exposure at 285°F (141°C)

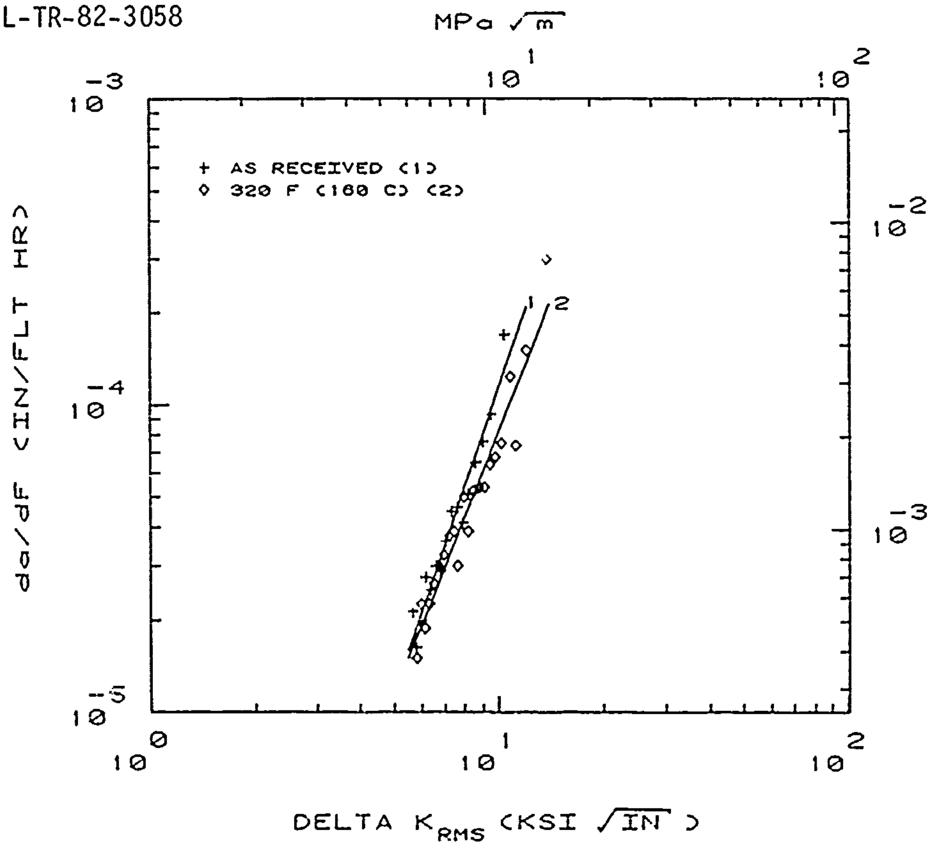


Figure 5. Crack Growth Rates After Exposure at 320°F (160°C)

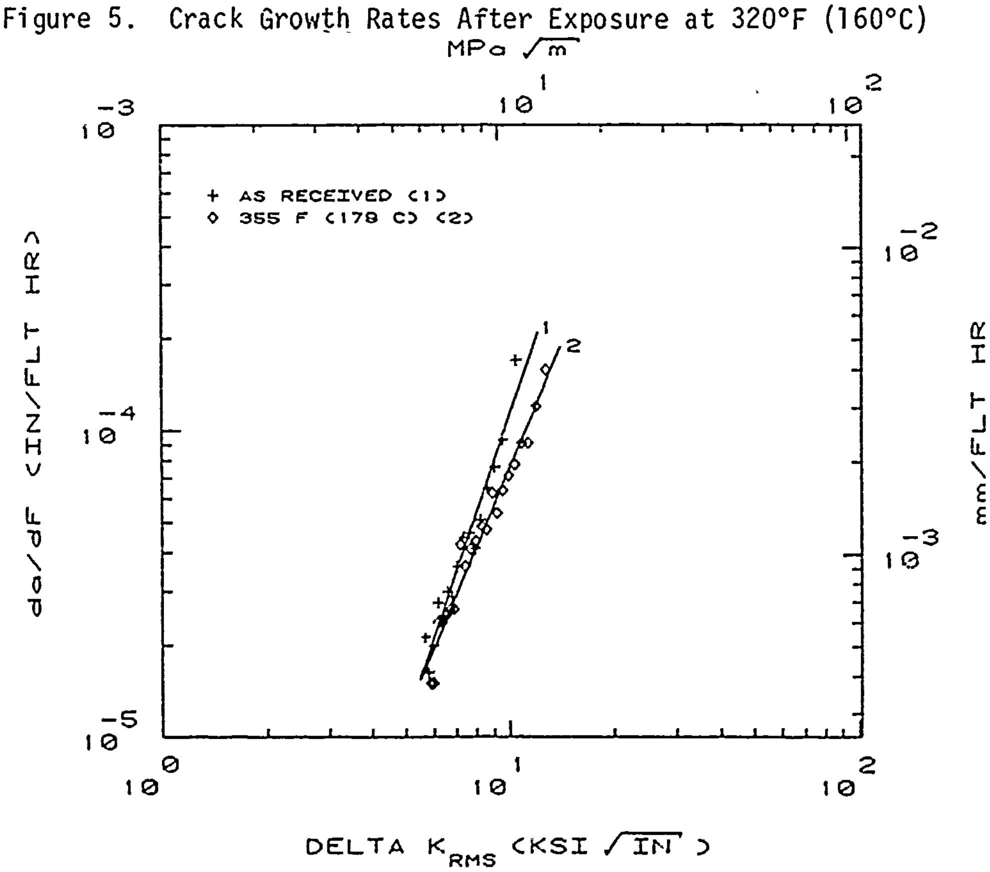


Figure 6. Crack Growth Rates After Exposure at 355°F (179°C)

material property. Data from these tests are included in Appendix C. Toughness and percent elongation were not evaluated. All results were normalized to values obtained from the baseline (as received) material and plotted as a function of exposure temperature. Figure 7 shows the observed relationship between these properties and crack growth life. It's obvious that elevated temperatures affect specimen life much more than they affect other characteristics.

5. LOAD HISTORY DEPENDENCE

Since baseline data were already available from other tests being run in the laboratory, a "worst case" exposure was tested under constant amplitude loading. A center-cracked panel identical to two others being tested (7075-T651 aluminum, cross section of 0.25" x 3.95" -- 6.35mm x 100.33mm) was heated at 355°F (179°C) for two hours and inserted into the constant amplitude test matrix. The specimen was then tested under the same conditions as the other two specimens. The maximum applied stress was 9.9 Ksi (68 MPa) and the stress ratio (R) was 0.5. Loads were applied at an average rate of 1 Hz. The effects of heat exposure were hardly noticeable under these loading conditions. Crack growth rates were essentially unchanged and the difference in specimen lives was only 10 to 12% (Figure 8).

The disagreement between constant amplitude and flight-by-flight test results led to an evaluation of the microstructural changes associated with short-term thermal cycle. Specimen microstructure was evaluated using differential scanning calorimetry, and the observed changes can generally be described as overaging. Results were consistent with previous work (Reference 11) involving 10 minute heat treatments at 310°F (155°C) and higher.

In a paper which specifically discusses the ranking of fatigue crack growth resistance of 7000 series alloys, Bucci et al. separate precipitate microstructure effects into two categories depending upon the load spectrum (Reference 12). For spectra with low or infrequent overloads, overaging is expected to decrease crack growth rate. For

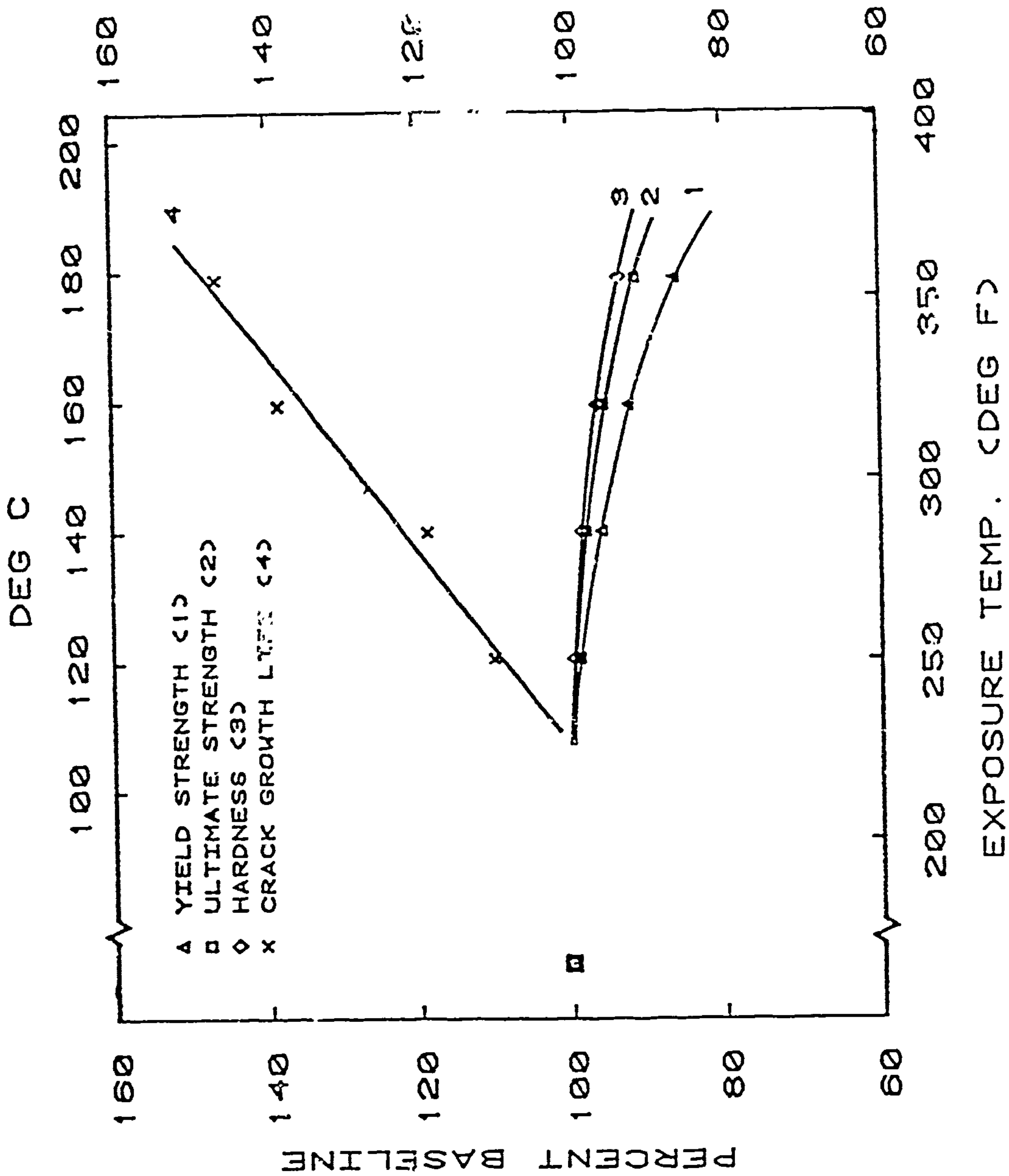


Figure 7. Relationship of Specimen Life to Material Properties

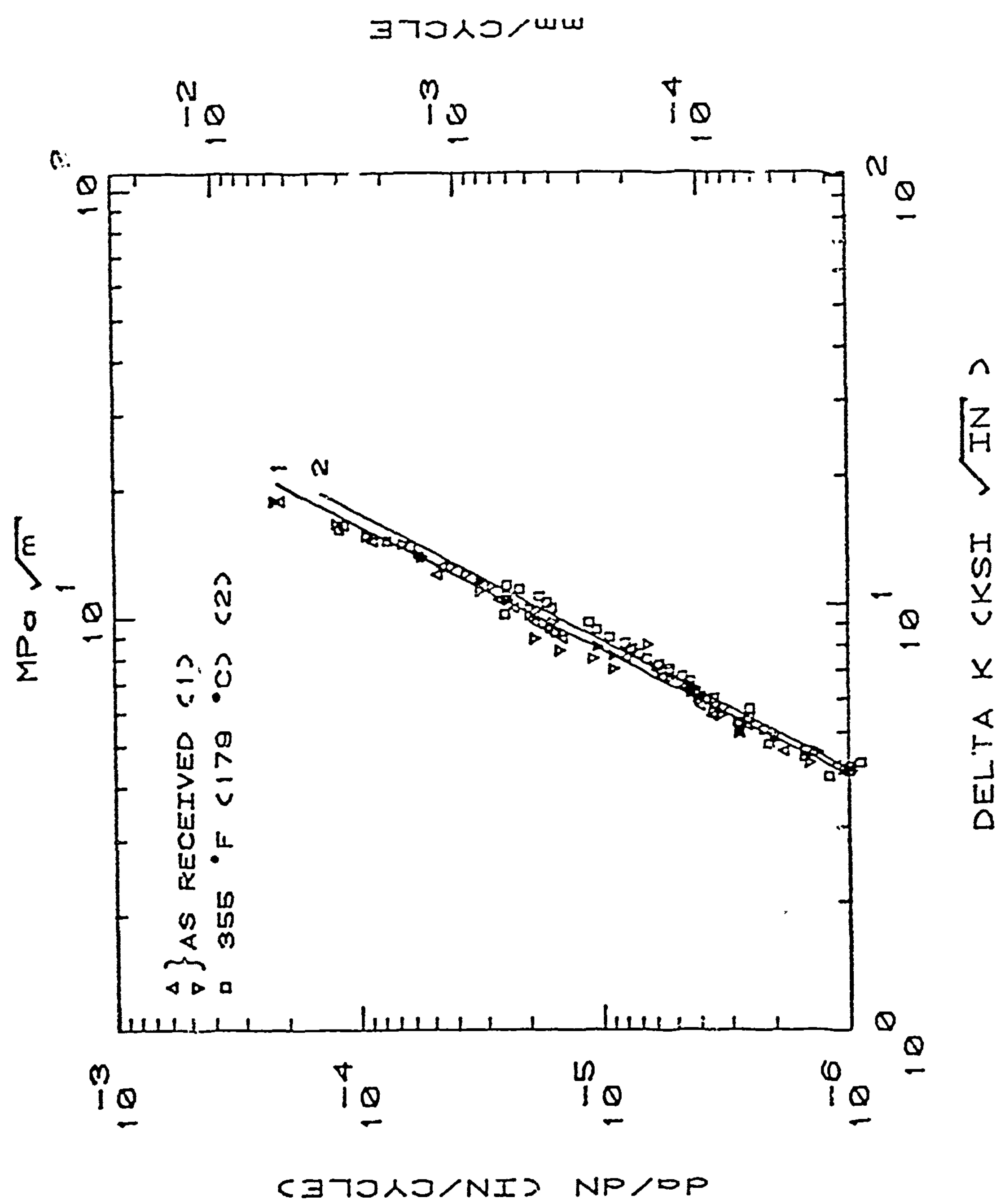


Figure 8. Crack Growth [illegible]s Observed Under Constant Amplitude Loading

spectra with high overloads, the effects of overaging on crack growth rate change with the mean stress intensity, and no overall prediction is possible. The importance of the load sequence and intensity on crack growth resistance was further illustrated in this reference by showing that 7075-T7 had better crack growth resistance than T6 in constant amplitude tests, but in a periodic spike overload test with an overload ratio of 1.8 and an occurrence of 1 in 4000 the T6 was far better than T7. For other overload ratios and occurrences the T7 was better. In summary, the detailed loading history can have profound effects on the relative fatigue resistance of various precipitate microstructures, and accurate predictions are not currently possible.

SECTION IV

SUMMARY OF HEAT EFFECTS

Exposure to temperatures of 200°F (93°C) or less did not appear to affect specimen life. However, temperatures above the minimum aging temperature of 240°F (116°C) produced a marked increase in life. The relationship appears to be somewhat linear, with a 100°F (56°C) increase in exposure temperature resulting in a 35-40% increase in crack growth life.

Observations made above were based on a least squares linear regression analysis of 16 data points. The resulting equation was

$$\%\ \text{Baseline Life} = 100 + 0.37\ (T-226) \qquad (2)$$

where T is exposure temperature in °F, and T is greater than 226°F (108°C).

Remember, this equation was derived from flight-by-flight loading conditions. The magnitude of observed temperature effects has been shown to depend on the type and severity of loading experienced after exposure.

SECTION V

MIL-HDBK-5C GUIDELINES FOR STRENGTH

After tensile and ultimate strengths were measured, test results were compared with design guidelines published in MIL-HDBK-5C. Values obtained from test specimens during this program were generally lower than those predicted by the handbook. Figures 9-12 show the results of these comparisons.

Although the amount of tensile data generated under this program is not statistically significant, it does show a need for caution. The curves presented in Figures 3.7.3.1.1 (a) and (b) of MIL-HDBK-5C are reproduced in Appendix C. These curves were developed using the rate process theory with the Larson-Miller time-temperature parameter $T(c + \log t)$. Data used to develop these curves were generated prior to 1960. Given the inherent limitations of analytical models, and that production techniques have changed since the models were verified, the prediction is suspect for some exposure conditions.

In situations where strength must be known, such as sweat-fitting bushings into lugs, the MIL-HDBK-5C curves should be used with caution. Tensile yield and ultimate strengths measured during this program decreased more than indicated by the handbook. Experimental verification is recommended in lieu of using these curves.

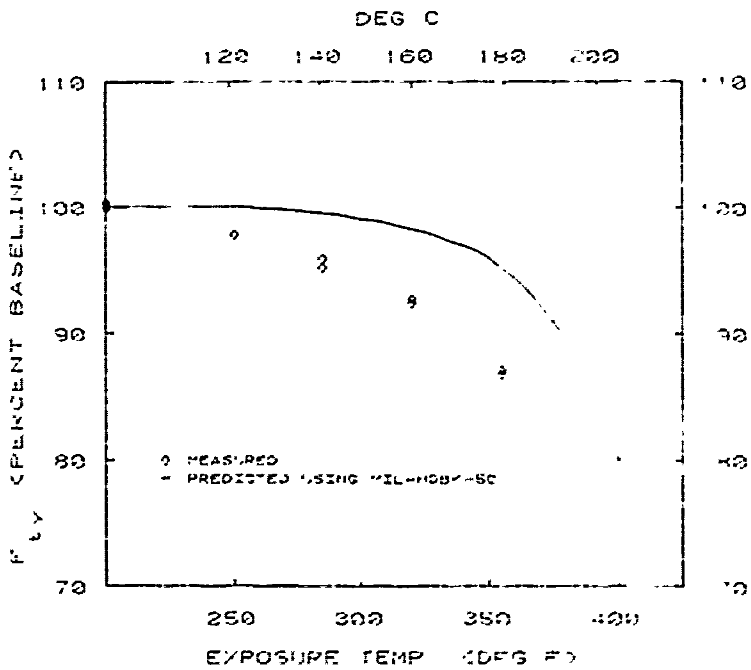


Figure 9. Yield Strength After One Hour Exposure: Data vs MIL-HDBK-5C Predictions

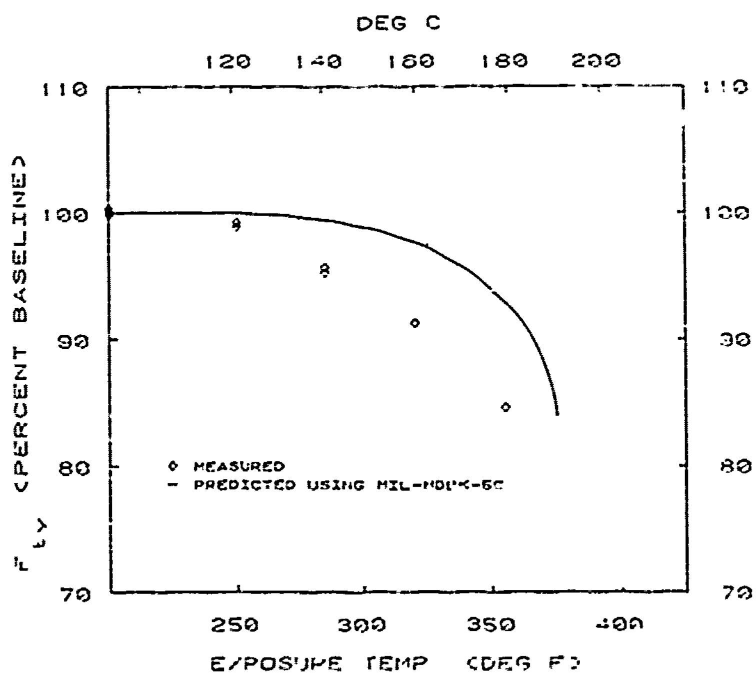


Figure 10. Yield Strength After Two Hour Exposure: Data vs MIL-HDBK-5C Predictions

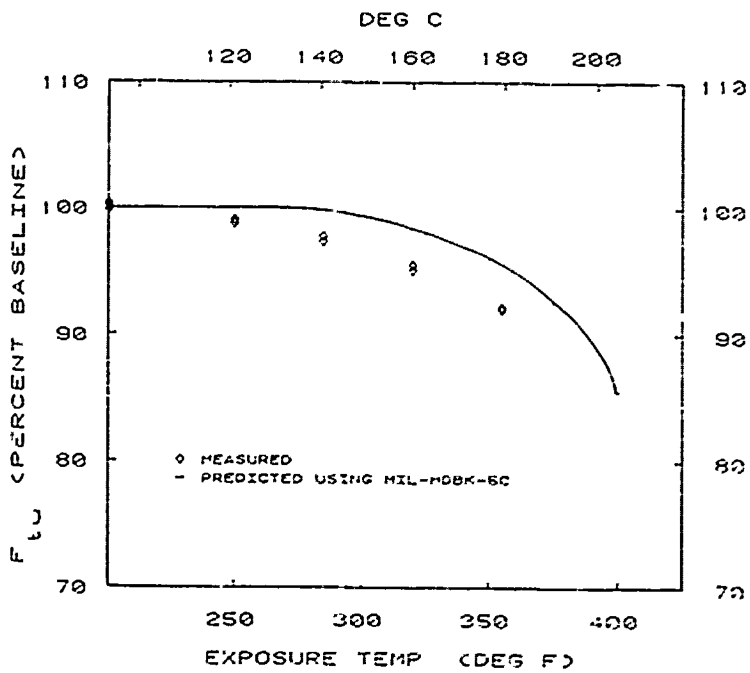


Figure 11. Ultimate Strength After One Hour Exposure: Data vs MIL-HDBK-5C Predictions

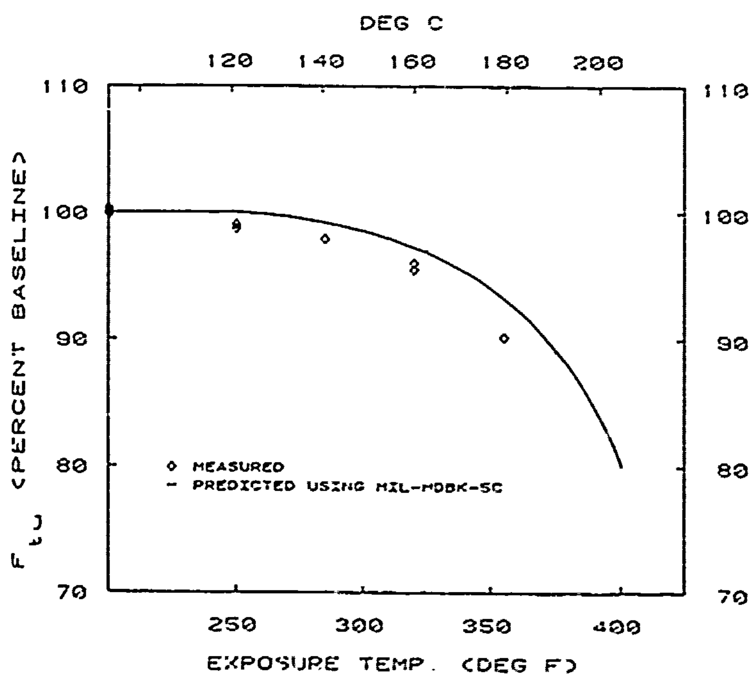


Figure 12. Ultimate Strength After Two Hour Exposure: Data vs MIL-HDBK-5C Predictions

SECTION VI

IMPLICATIONS FOR THE CRACK GROWTH GAGE

The crack growth gage is intended to be the primary structural monitoring device for USAF fighter, attack, and trainer aircraft. As such, it must stand alone and provide reliable, consistent data. An unexpected change in crack growth gage life could lead to excessive maintenance costs or the loss of aircraft which are thought to be "safe".

In the laboratory, crack growth gages can be successfully bonded to carrier specimens using temperatures of 200 to 225°F (93 to 107°C). However, overcoming the heatsink effects of a large aircraft wing structure is much more difficult than placing a coupon in an oven. References 3 and 5 found that control temperatures in excess of 300°F (149°C) were required to obtain an acceptable bond using heat blankets and vacuum bags.

The use of heat-cure adhesives to bond crack growth gages to an aircraft wingskin requires a great deal of caution. If temperatures above 225°F (107°C) are applied during the bonding process, their effects on the crack growth behavior of the gage material must be understood for all projected loading conditions. Since crack growth gages are designed to experience higher stress levels than the host structure, the magnitude of observed heat effects may vary, depending on mission profile and gage-to-structure stress ratio. Until these effects are fully understood, crack growth cannot even be predicted for the gage itself; certainly it cannot be predicted for the structure.

SECTION VII

CONCLUSIONS

1. Under flight-by-flight loading conditions, the crack growth lives of 7075-T651 aluminum specimens which had been exposed to temperatures between 250 and 355°F (121 and 179°C) were consistently longer than the lives of baseline specimens.

2. Differences in test results for one hour and two hour exposure times were negligible.

3. Data from constant amplitude tests did not support the trend which was observed under flight-by-flight loading. The type and severity of loading experienced after exposure influenced specimen response.

4. As expected, exposure to elevated temperatures caused specimen yield strength, ultimate strength, and hardness to decrease.

5. Data for tensile yield and ultimate strengths generated under this program did not agree with the design curves published in MIL-HDBK-5C. The curves appear to be unconservative.

SECTION VIII

RECOMMENDATIONS

1. Heat-cure adhesives should not be used to bond crack growth gages to aircraft components at temperatures above 225°F (107°C).

2. Figures 3.7.3.1.1 (a) and (b) of MIL-HDBK-5C should be used with caution. Experimental verification is recommended in lieu of these figures.

APPENDIX A

LOAD HISTORY

The load history used for this program was applied as a blocked flight-by-flight history and was repeated every 400 equivalent flight hours. Twenty repetitions (units) comprised a lifetime of 8000 flight hours.

Table A-1 gives a block-by-block breakdown of the history. All values are listed as percent design stress (100% = 29 Ksi). The number of repetitions for a particular load level, however, may vary from unit to unit. For example, load level number 22 occurs 1.2 times. This means that the load is applied once (1) during each repetition of the load history plus one additional time for every fifth repetition (.2 = 1/5) of the unit history. This load level would occur 24 times during one lifetime. Exceedance curves for the peak (maximum) and range (maximum minus minimum) load levels are shown in Figure A-1. Figures A-2 and A-3 present occurrences by load level.

Root-mean-square stresses calculated for the load history were:

RMS Maximum Stress	11.855 Ksi
RMS Minimum Stress	8.446 Ksi
RMS Delta Stress	3.659 Ksi

Table A-1: F-16 Lower Wingskin Load History
(Compressive Loads Clipped at Zero)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
15.30	53.70	10.00	1
10.00	25.00	3.00	2
15.30	47.80	22.00	3
10.00	16.10	1.00	4
6.70	29.70	64.00	5
11.90	51.00	11.00	6
14.20	37.10	49.00	7
14.50	40.20	.10	8
11.50	41.00	.50	9
13.40	27.10	1.00	10
6.30	17.80	1.00	11
11.50	53.80	10.00	12
11.50	41.30	27.00	13
9.60	27.00	18.00	14
6.90	14.50	1.00	15
7.30	18.80	1.00	16
11.50	26.30	76.00	17
0.00	6.90	1.00	18
0.00	92.30	9.00	19
13.40	31.80	1308.00	20
11.50	35.00	49.00	21
0.00	100.00	1.20	22
13.40	49.30	618.00	23
13.80	44.40	38.00	24
15.30	53.70	11.00	25
15.30	51.00	302.00	26
14.90	95.30	1.40	27
12.40	40.60	305.00	28
11.90	44.20	95.00	29
6.10	15.30	1.00	30
13.40	43.60	3.00	31
15.30	33.70	77.00	32
0.00	11.90	1.00	33
14.50	40.20	15.00	34
13.40	26.10	1.00	35
11.50	34.50	.10	36
9.60	37.90	5.00	37
15.30	56.30	32.00	38
0.00	15.30	1.00	39
13.40	41.30	1129.00	40

Table A-1 (Continued)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
14.90	50.50	43.00	41
0.00	54.90	80.00	42
10.40	16.80	1.00	43
15.30	54.40	.20	44
0.00	13.40	1.00	45
15.30	37.50	350.00	46
15.30	64.90	27.00	47
11.50	38.80	1.00	48
0.00	83.50	30.00	49
15.30	51.00	7.00	50
12.40	51.80	.05	51
13.40	38.90	6.00	52
7.10	13.50	1.00	53
13.80	33.00	321.00	54
0.00	14.20	1.00	55
13.40	51.50	211.00	56
11.90	80.80	.50	57
3.10	11.90	1.00	58
14.90	36.60	54.00	59
11.50	43.30	5.00	60
15.30	55.20	9.00	61
11.90	37.10	15.00	62
11.90	30.10	57.00	63
11.90	44.20	1.00	64
15.30	59.00	89.00	65
11.90	64.40	3.00	66
15.30	58.10	.50	67
9.60	59.80	13.00	68
15.30	47.80	43.00	69
11.50	41.90	3.00	70
11.90	64.40	4.00	71
0.00	57.80	21.00	72
15.30	21.40	1.00	73
0.00	9.60	1.00	74
15.70	65.00	3.00	75
0.00	15.30	6.00	76
11.50	28.00	57.00	77
6.90	25.70	56.00	78
13.40	53.10	591.00	79
9.60	14.50	1.00	80

Table A-1 (Continued)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
9.40	20.90	11.00	81
15.30	33.70	347.00	82
11.90	64.40	1.00	83
11.50	36.60	98.00	84
10.00	29.70	880.00	85
11.90	67.80	28.00	86
13.40	37.50	1.00	87
15.30	33.70	74.00	88
11.50	34.50	5.00	89
9.60	27.10	52.00	90
0.00	59.00	13.00	91
12.70	19.10	1.00	92
12.40	48.60	.10	93
11.50	35.00	8.00	94
11.90	44.00	44.00	95
14.50	20.70	1.00	96
5.60	21.70	99.00	97
9.60	49.30	1.00	98
14.50	35.60	42.00	99
11.90	77.50	.05	100
0.00	11.90	1.00	101
10.00	72.00	80.00	102
14.90	63.30	22.00	103
0.00	67.60	6.00	104
10.00	54.50	416.00	105
15.30	55.20	2.00	106
12.40	17.60	1.00	107
10.50	16.90	1.00	108
11.90	18.30	1.00	109
11.90	30.10	347.00	110
0.00	82.30	2.00	111
15.30	79.20	14.00	112
11.90	57.70	7.00	113
11.90	71.60	2.00	114
14.90	77.50	6.00	115
13.80	79.30	3.00	116
13.40	31.60	24.00	117
11.90	57.70	2.00	118
13.40	29.90	6.00	119
12.40	53.50	1.00	120

Table A-1 (Continued)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
8.80	15.30	1.00	121
11.50	34.50	876.00	122
13.80	70.30	23.00	123
14.50	35.60	125.00	124
4.20	14.20	1.00	125
4.00	15.50	1.00	126
13.40	30.90	156.00	127
10.00	42.10	760.00	128
6.90	37.70	48.00	129
12.40	33.30	5.00	130
11.50	30.30	4.00	131
0.00	28.70	3.00	132
15.30	47.80	8.00	133
13.40	50.50	1.00	134
13.40	31.80	3.00	135
14.50	40.20	29.00	136
13.80	44.60	1082.00	137
13.40	49.50	1.00	138
15.30	56.30	2.00	139
0.00	11.90	1.00	140
6.90	49.70	27.00	141
11.90	44.00	6.00	142
9.60	37.90	45.00	143
0.00	9.60	1.00	144
14.20	34.30	57.00	145
11.90	56.90	593.00	146
0.00	15.30	17.00	147
11.50	16.90	1.00	148
0.00	11.90	1.00	149
11.50	30.30	19.00	150
13.80	67.60	153.00	151
0.00	78.30	4.00	152
10.10	26.20	189.00	153
11.50	33.30	22.00	154
7.30	13.80	1.00	155
0.00	11.90	1.00	156
11.90	68.50	10.00	157
0.00	14.20	1.00	158
15.70	37.70	16.00	159
11.90	37.10	36.00	160

Table A-1 (Continued)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
11.90	71.60	.50	161
0.00	11.50	1.00	162
12.40	45.50	2.00	163
13.40	27.10	1.00	164
13.40	48.40	2.00	165
15.30	55.20	7.00	166
1.10	24.10	18.00	167
11.50	28.00	43.00	168
2.30	18.40	11.00	169
11.90	31.60	4.00	170
0.00	18.80	5.00	171
6.90	68.00	5.00	172
11.50	41.00	1.00	173
12.40	30.30	885.00	174
15.30	56.90	1.00	175
13.40	37.50	.10	176
15.30	40.60	36.00	177
11.10	22.60	1.00	178
12.40	47.10	30.00	179
13.40	28.20	17.00	180
13.40	30.90	12.00	181
15.30	40.60	87.00	182
11.50	39.70	36.00	183
13.40	34.10	15.00	184
15.30	56.00	3.00	185
14.50	48.20	.20	186
15.70	65.00	5.00	187
13.80	33.00	152.00	188
15.30	56.90	3.00	189
0.00	6.90	1.00	190
13.80	33.00	1353.00	191
6.90	23.00	11.00	192
13.80	53.90	7.00	193
9.60	63.90	6.00	194
13.40	41.30	8.00	195
15.30	37.50	122.00	196
10.00	67.00	224.00	197
13.40	52.20	1.00	198
11.90	57.70	9.00	199
15.30	75.80	5.00	200

Table A-1 (Concluded)

MINIMUM STRESS	MAXIMUM STRESS	NUMBER OF CYCLES	
15.30	51.00	34.00	201
9.60	49.30	25.00	202
0.00	60.80	3.00	203
13.40	19.10	1.00	204
15.30	56.00	138.00	205
0.00	6.90	1.00	206
13.80	44.40	3.00	207
13.40	34.50	29.00	208
13.40	29.30	57.00	209
0.00	14.20	1.00	210
2.10	34.30	4.00	211
6.90	61.90	14.00	212
15.70	51.40	17.00	213
13.40	33.70	2.00	214
12.40	27.40	30.00	215
14.20	40.40	47.00	216
11.90	71.60	4.00	217
13.40	41.30	52.00	218
0.00	15.30	57.00	219
15.70	37.50	69.00	220
11.90	30.10	76.00	221
11.50	44.20	14.00	222
4.40	15.90	1.00	223
0.00	13.40	1.00	224
2.70	11.90	1.00	225
11.50	38.80	479.00	226
15.70	50.70	5.00	227
11.90	44.00	22.00	228
13.40	16.90	1.00	229
12.40	39.20	2.00	230
15.30	58.10	6.00	231
11.90	51.00	26.00	232
15.30	58.10	3.00	233
0.00	11.50	1.00	234
11.50	30.30	1014.00	235
11.90	32.00	109.00	236
14.50	44.40	6.00	237
15.30	53.70	26.00	238
3.10	19.20	7.00	239
0.00	43.50	1.00	240

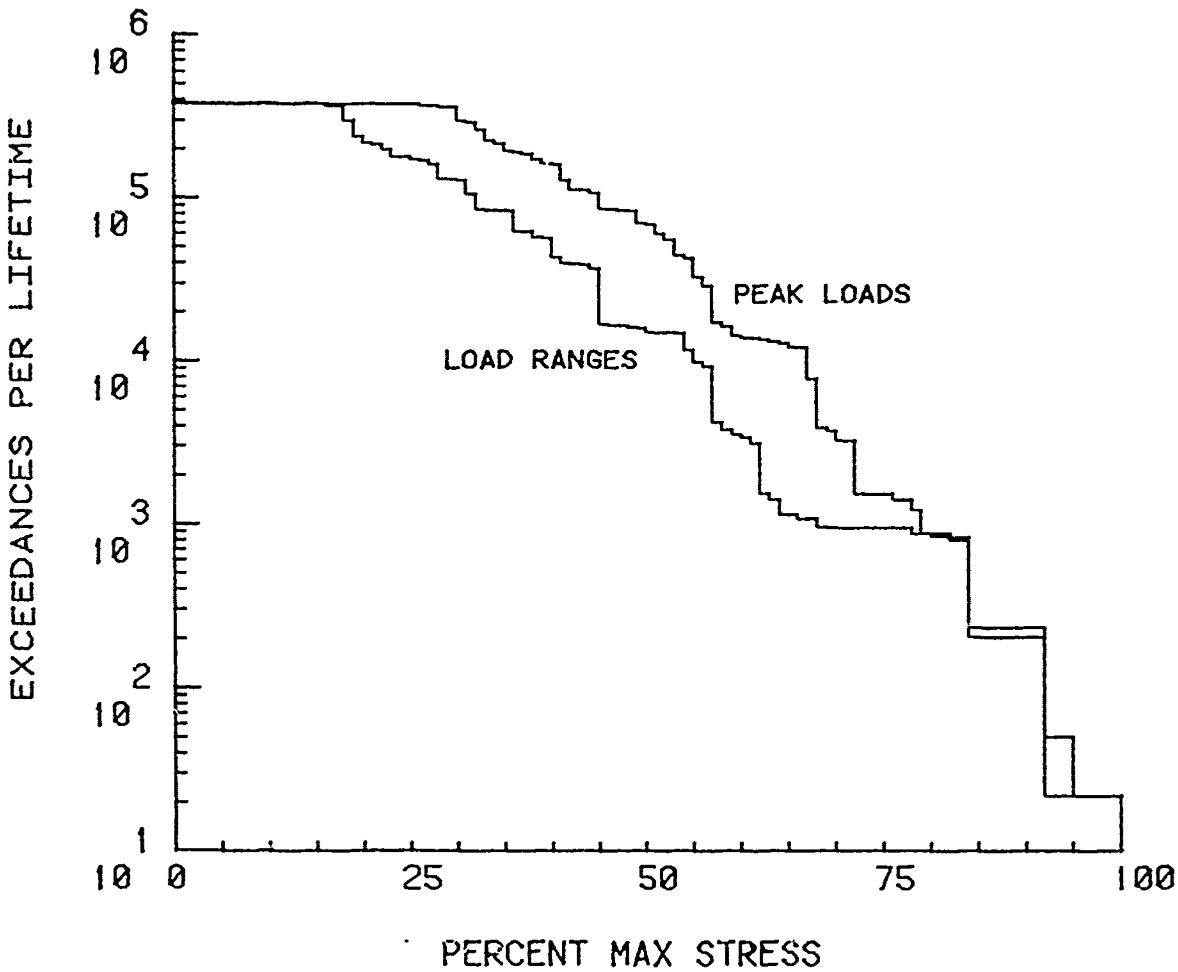


Figure A-1. Load History Exceedance Curves

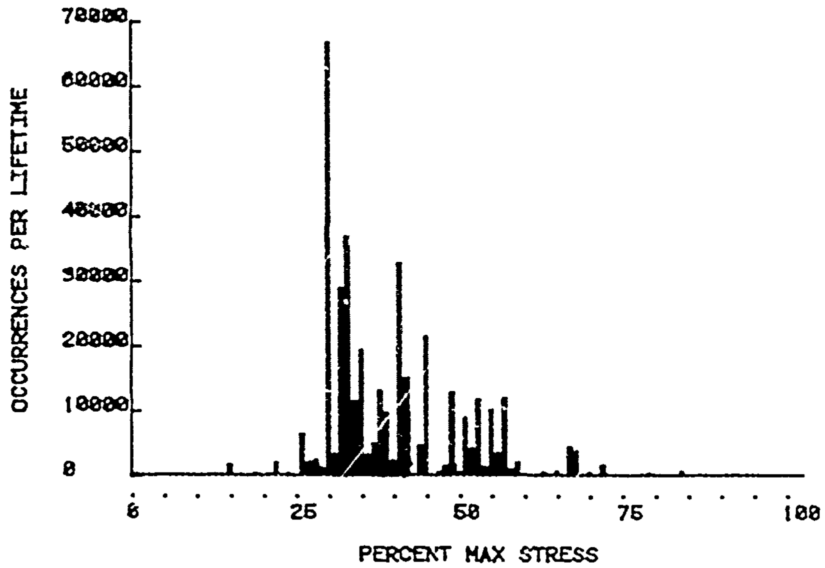


Figure A-2. Occurrence Histogram for Peak Loads

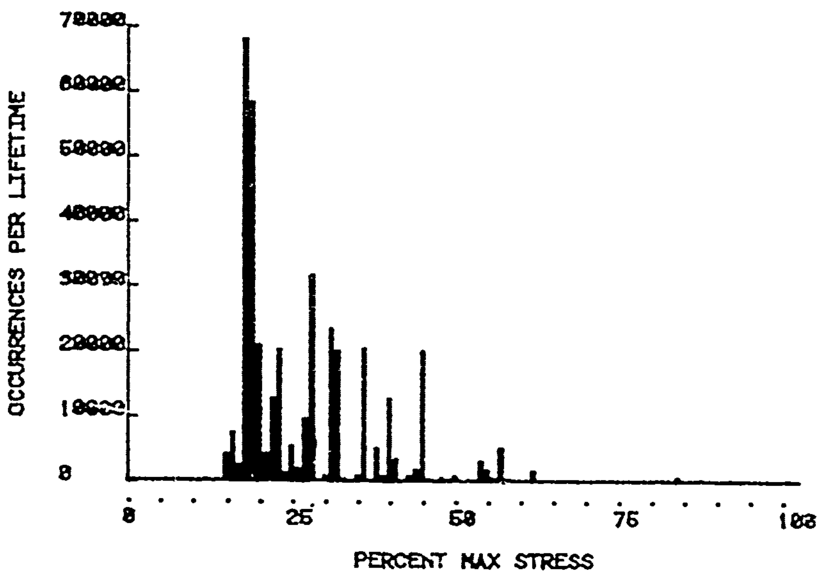


Figure A-3. Occurrence Histogram for Load Ranges

APPENDIX B

DATA LISTS

Data for all specimens were collected in the form of total crack length (2a, in inches) vs. time (in equivalent flight hours). The next 10 pages show raw data by temperature group.

The crack growth life of each specimen was computed by interpolating the number of flight hours required for the total crack length to reach 0.3 inches and subtracting that value from the number of flight hours accumulated at failure. For example, specimen B-5 had a total crack length of 0.286 inches after 2000 flight hours, and a length of 0.304 inches after 2400 flight hours. The value for F(0.3) was computed as follows:

$$\frac{F(0.3) - 2000}{2400 - 2000} = \frac{0.3 - 0.286}{0.304 - 0.286}$$

This yielded a value of 2311 flight hours. Specimen B-5 failed after 8039 flight hours of testing, so the crack growth life was

$$8039 - 2311 = 5728 \text{ Flt hrs}$$

Specimen lives are listed in Table B-1.

BASELINE

FLT HRS	B-5	B-6	B-7	B-8	AVERAGE
0	0.253	0.255	0.252	0.252	0.253
400	0.253	0.255	0.252	0.252	0.253
800	0.259	0.255	0.259	0.257	0.258
1200	0.265	0.255	0.263	0.261	0.261
1600	0.269	0.255	0.269	0.264	0.264
2000	0.286	0.278	0.275	0.285	0.281
2400	0.304	0.286	0.285	0.301	0.294
2800	0.326	0.298	0.294	0.320	0.310
3200	0.356	0.319	0.309	0.343	0.332
3600	0.376	0.338	0.330	0.365	0.352
4000	0.407	0.354	0.355	0.388	0.376
4400	0.435	0.376	0.375	0.410	0.399
4800	0.469	0.408	0.400	0.433	0.428
5200	0.505	0.441	0.439	0.471	0.464
5600	0.554	0.476	0.468	0.507	0.501
6000	0.600	0.501	0.489	0.546	0.534
6400	0.640	0.539	0.525	0.596	0.575
6800	0.713	0.577	0.559	0.660	0.627
7200	0.766	0.641	0.614	0.732	0.688
7600	0.879	0.702	0.657	0.814	0.763
8000	1.145	0.780	0.695	0.977	0.899
8400			0.734	1.279	1.007
8800		1.094	0.799		
9200			0.869		
9600			0.991		
10000			1.225		

355°F (179°C)

FLT HRS	1A-1	1A-2	1B-1	1B-2	AVERAGE
0	0.248	0.256	0.250	0.252	0.252
400	0.248	0.256	0.250	0.256	0.253
800	0.261	0.262	0.262	0.260	0.261
1200	0.267	0.268	0.266	0.263	0.266
1600	0.270	0.274	0.269	0.274	0.272
2000	0.278	0.282	0.279	0.280	0.280
2400	0.283	0.295	0.288	0.286	0.288
2800	0.302	0.303	0.299	0.296	0.300
3200	0.315	0.314	0.308	0.312	0.312
3600	0.333	0.330	0.335	0.324	0.331
4000	0.354	0.349	0.357	0.339	0.350
4400	0.382	0.363	0.377	0.358	0.370
4800	0.402	0.388	0.394	0.379	0.391
5200	0.410	0.406	0.432	0.401	0.412
5600	0.461	0.437	0.464	0.421	0.446
6000	0.492	0.465	0.495	0.449	0.475
6400	0.529	0.494	0.531	0.477	0.508
6800	0.573	0.524	0.570	0.506	0.543
7200	0.616	0.563	0.605	0.542	0.582
7600	0.665	0.579	0.656	0.581	0.620
8000	0.714	0.645	0.703	0.617	0.670
8400	0.765	0.679	0.748	0.658	0.713
8800	0.815	0.729	0.806	0.704	0.764
9200	0.889	0.779	0.868	0.748	0.821
9600	0.961	0.830	0.933	0.806	0.883
10000	1.050	0.885	1.019	0.868	0.956
10400	1.140	0.939	1.112	0.924	1.029
10800	1.264	1.003	1.241	0.990	1.125
11200	1.440	1.075	1.431	1.062	1.252
11600		1.164		1.174	
12000		1.275		1.341	
12400		1.410		1.628	
12800		1.706			

320°F (160°C)

FLT HRS	2A-1	2A-2	2B-1	2B-2	AVERAGE
0	0.251	0.252	0.248	0.249	0.250
400	0.251	0.252	0.248	0.249	0.250
800	0.257	0.259	0.248	0.259	0.256
1200	0.258	0.265	0.250	0.265	0.260
1600	0.265	0.271	0.267	0.270	0.268
2000	0.272	0.280	0.276	0.275	0.276
2400	0.285	0.282	0.283	0.281	0.283
2800	0.298	0.291	0.298	0.293	0.295
3200	0.315	0.309	0.316	0.310	0.313
3600	0.333	0.322	0.335	0.321	0.328
4000	0.356	0.342	0.354	0.332	0.346
4400	0.376	0.358	0.380	0.354	0.367
4800	0.404	0.381	0.410	0.370	0.391
5200	0.430	0.402	0.440	0.397	0.417
5600	0.458	0.430	0.475	0.423	0.447
6000	0.496	0.470	0.499	0.448	0.478
6400	0.521	0.485	0.520	0.482	0.502
6800	0.549	0.547	0.561	0.510	0.542
7200	0.591	0.561	0.603	0.538	0.573
7600	0.626	0.591	0.674	0.569	0.615
8000	0.677	0.630	0.715	0.608	0.658
8400	0.724	0.666	0.764	0.650	0.701
8800	0.781	0.701	0.828	0.696	0.752
9200	0.840	0.751	0.890	0.742	0.806
9600	0.908	0.792	0.968	0.794	0.866
10000	0.979	—	1.069	0.847	0.965
10400	1.059	0.922	1.209	0.905	1.024
10800	1.170	0.949	1.491	0.970	1.145
11200	1.376	1.038		1.049	1.154
11600		1.140		1.200	1.170
12000		1.275		1.544	1.410
12400		1.520			

$285^{o}F$ ($141^{o}C$)

FLT HRS	3A-2	3A-3	3B-1	3B-2	AVERAGE
0	0.253	0.250	0.248	0.251	0.251
400	0.253	0.250	0.248	0.251	0.251
800	0.254	0.253	0.252	0.257	0.254
1200	0.267	0.266	0.263	0.265	0.265
1600	0.279	0.280	0.273	0.270	0.276
2000	0.283	0.288	0.280	0.286	0.284
2400	0.284	0.316	0.289	0.295	0.296
2800	0.291	0.336	0.301	0.310	0.310
3200	0.309	0.360	0.322	0.327	0.330
3600	0.321	0.390	0.334	0.348	0.348
4000	—	0.420	0.356	0.373	0.383
4400	0.355	0.457	0.384	0.392	0.397
4800	0.388	0.505	0.418	0.417	0.432
5200	0.401	0.550	0.449	0.441	0.460
5600	0.425	0.604	0.482	0.471	0.496
6000	0.449	0.657	0.529	0.503	0.535
6400	0.475	0.703	0.569	0.526	0.568
6800	0.515	0.766	0.615	0.558	0.614
7200	0.558	0.832	0.654	0.604	0.662
7600	0.606	0.914	0.710	0.646	0.719
8000	0.647	1.020	0.759	0.678	0.776
8400	0.696	1.203	0.814	0.718	0.858
8800	0.759	1.580	0.890	0.766	0.999
9200	0.801		1.015	0.820	
9600	0.878		1.196	0.875	
10000	0.967			0.951	
10400	1.162			1.065	
10800				1.301	

285 DEGREES

FLT HRS	3A-1*
0	0.255
400	0.255
800	0.255
1200	0.258
1600	0.26
2000	0.267
2400	0.295
2800	0.309
3200	0.333
3600	0.356
4000	0.381
4400	0.416
4800	0.449
5200	0.486
5600	0.52
6000	0.55
6400	0.588
6800	0.623
7200	0.666
7600	0.71
8000	0.773

* SPECIMEN ACCIDENTALLY OVERLOADED AFTER 8000 FLT HRS
DATA COLLECTION TERMINATED AT OVERLOAD

250°F (121°C)

FLT HRS	4A-1	4A-2	4B-1	4B-2	AVERAGE
0	0.250	0.250	0.248	0.251	0.250
400	0.250	0.250	0.248	0.251	0.250
800	0.256	0.250	0.254	0.251	0.253
1200	0.259	0.255	—	0.265	0.260
1600	0.278	0.271	0.272	0.268	0.272
2000	0.291	0.276	0.273	0.275	0.279
2400	0.328	0.287	0.287	0.279	0.295
2800	0.339	0.296	0.293	0.294	0.306
3200	0.362	0.310	0.310	0.308	0.323
3600	0.380	0.330	0.334	0.329	0.343
4000	0.419	0.350	0.360	0.347	0.369
4400	0.451	0.376	0.376	0.367	0.393
4800	0.507	0.415	0.399	0.390	0.428
5200	0.540	0.450	0.424	0.414	0.457
5600	0.579	0.489	0.456	0.446	0.493
6000	0.635	0.520	0.481	0.470	0.527
6400	0.694	0.554	0.509	0.490	0.562
6800	0.737	0.593	0.541	0.531	0.601
7200	0.825	0.636	0.584	0.559	0.651
7600	0.895	0.687	0.623	0.591	0.699
8000	1.049	0.743	0.677	0.637	0.777
8400	1.607	0.788	0.709	0.680	0.946
8800		0.872	0.760	0.738	0.790
9200		0.965	0.907	0.795	0.889
9600		1.124	1.105	0.872	1.034
10000		1.543	1.698	1.003	1.415
10400				1.246	

250°F (121°C)

FLT HRS	4C-1	4C-2	AVERAGE
0	0.251	0.250	0.251
400	0.251	0.250	0.251
800	0.251	0.250	0.251
1200	0.253	0.262	0.258
1600	0.265	0.262	0.264
2000	0.279	0.271	0.275
2400	0.288	0.280	0.284
2800	0.305	0.292	0.299
3200	0.319	0.304	0.312
3600	0.335	0.322	0.329
4000	0.352	0.340	0.346
4400	0.373	0.358	0.366
4800	0.395	0.370	0.383
5200	0.419	0.391	0.405
5600	0.441	0.414	0.428
6000	0.461	0.434	0.448
6400	0.501	0.478	0.490
6800	0.541	0.500	0.521
7200	0.578	0.527	0.553
7600	0.629	0.555	0.592
8000	0.658	0.597	0.628
8400	0.726	0.640	0.683
8800	0.780	0.680	0.730
9200	0.857	0.715	0.786
9600	0.960	0.766	0.863
10000	1.163	0.825	0.994
10400		0.912	
10800		1.076	
11200		1.489	

235°F (113°C)

FLT HRS	6B-1	6B-2	AVERAGE
0	0.251	0.252	0.252
400	0.251	0.252	0.252
800	0.251	0.252	0.252
1200	0.261	0.260	0.261
1600	0.261	0.265	0.263
2000	0.264	0.275	0.270
2400	0.291	0.283	0.287
2800	0.305	0.295	0.300
3200	0.320	0.308	0.314
3600	0.340	0.319	0.330
4000	0.363	0.332	0.348
4400	0.384	0.361	0.373
4800	0.405	0.375	0.390
5200	0.431	0.397	0.414
5600	0.455	0.432	0.444
6000	0.482	0.455	0.469
6400	0.531	0.486	0.509
6800	0.553	0.518	0.536
7200	0.582	0.551	0.567
7600	0.621	0.595	0.608
8000	0.665	0.636	0.651
8400	0.712	0.677	0.695
8800	0.758	0.716	0.737
9200	0.830	0.766	0.798
9600	0.950	0.837	0.894
10000	1.166	0.960	1.063
10400		1.171	

$200^{\circ}F$ ($93^{\circ}C$)

FLTHRS	8B-1
0	0.247
400	0.247
800	0.247
1200	0.258
1600	0.273
2000	0.285
2400	0.296
2800	0.31
3200	0.329
3600	0.347
4000	0.367
4400	0.395
4800	0.42
5200	0.446
5600	0.478
6000	0.515
6400	0.53
6800	0.567
7200	0.609
7600	0.665
8000	0.712
8400	0.785
8800	0.938
9200	1.08
9600	1.513

150°F (66°C)

FLT HRS	9B-1	9B-2	AVERAGE
0	0.249	0.258	0.254
400	0.249	0.258	0.254
800	0.249	0.261	0.255
1200	0.259	0.263	0.261
1600	0.275	0.265	0.270
2000	0.285	0.270	0.278
2400	0.298	0.278	0.288
2800	0.315	0.285	0.300
3200	0.334	0.297	0.316
3600	0.356	0.311	0.334
4000	0.381	0.331	0.356
4400	0.409	0.341	0.375
4800	0.450	0.371	0.411
5200	0.484	0.375	0.430
5600	0.510	0.402	0.456
6000	0.556	0.426	0.491
6400	0.587	0.455	0.521
6800	0.632	0.490	0.561
7200	0.677	0.530	0.604
7600	0.727	0.587	0.657
8000	0.791	0.629	0.710
8400	0.937	0.671	0.804
8800	1.180	0.724	0.952
9200		0.791	
9600		0.891	
10000		1.100	
10400		1.610	

TABLE B-1

SPECIMEN LIVES (Flight-By-Flight Loading)

SPECIMEN	HOURS TO 0.3 INCHES (INTERPOLATED)	HOURS TO FAILURE	CRACK GROWTH LIFE
B-5	2311	8039	5728
B-6	2838	9153	6315
B-7	2960	10284	7324
B-8	2375	8420	6045
1A-1	2758	11556	8798
1A-2	2650	12850	10200
1B-1	2844	11555	8711
1B-2	2900	12410	9510
2A-1	2847	11554	8707
2A-2	3000	12440	9440
2B-1	2844	10840	7996
2B-2	2965	12010	9045
3A-2	3000	10756	7756
3A-3	2171	8810	6639
3B-1	2767	9957	7190
3B-2	2533	11084	8551
4A-1	2097	8410	6313
4A-2	2914	10010	7096
4B-1	2965	10010	7045
4B-2	2971	10440	7469
4C-1	2682	10353	7671
4C-2	3067	11210	8143
6B-1	2657	10354	7697
6B-2	2954	10755	7801
8B-1	2514	9610	7096
9B-1	2447	9084	6637
9B-2	3286	10410	7124

APPENDIX C

MATERIAL PROPERTIES DATA

Tension test specimens (Figure C-1) were fabricated and tested in accordance with ASTM Standard E-8 (Reference 10). All specimens were tested in a 20 Kip Instron mechanical (screw-type) testing machine using a crosshead speed of 0.2 inches per minute. Data were automatically recorded on a strip chart. Strain gages placed on specimens 1 and 2 verified the accuracy of the recording system. Results of tension tests are shown in Table C-1.

Hardness data were also collected (Table C-2). Generally, only two readings were taken per specimen, but if these readings did not agree within 3 units on the Rockwell "B" scale, a third reading was taken. Specimen hardness was computed as the average of all readings.

MIL-HDBK-5C Values. Figures C-2 and C-3 show the expected tensile strengths of 7075-T6 aluminum alloys which have been exposed to elevated temperatures. (To determine the percent of "baseline" strength, locate exposure time on the right axis, move horizontally to intersect the appropriate exposure temperature, move vertically to intersect the testing temperature, then move horizontally to read percent F_{ty} or F_{tu} on the left axis.)

Yield and ultimate strength data obtained from tension tests did not agree with the MIL-HDBK-5C "expected values" for the exposures evaluated. For example, the expected yield strength after a one hour exposure at 350°F is 96% of the baseline value (Figure C-3). Empirical data (Table C-1) showed that after a one hour exposure at 355°F, specimen yield strength was only 87% of the baseline value.

NOTE: ALL DIMENSIONS
IN INCHES.

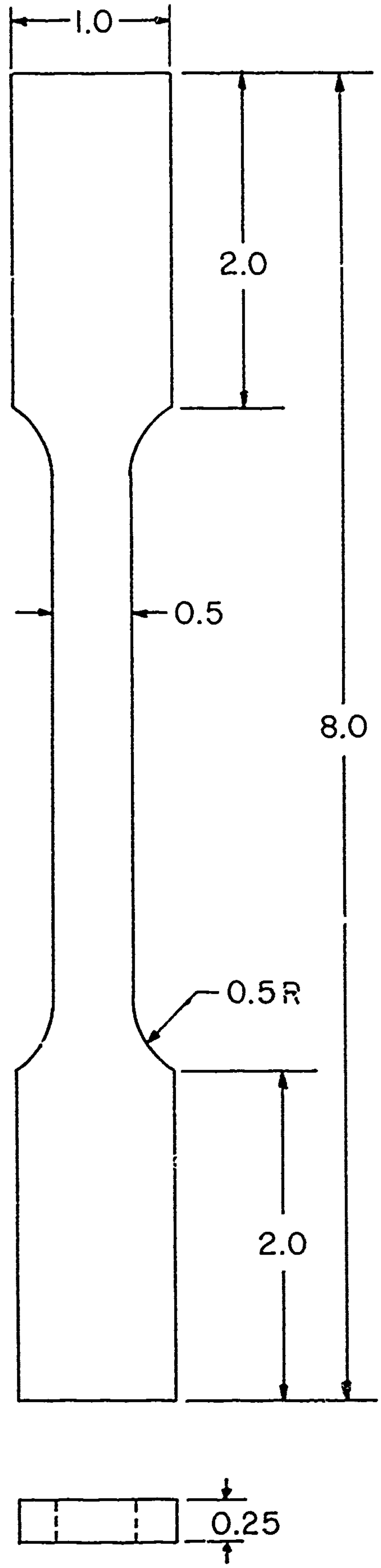


Figure C-1. Tension Test Specimen

TABLE C-1

TENSILE TEST RESULTS

TEST NUMBER	ORIGINAL SPECIMEN NUMBER	TREATMENT	$\sigma_{y0.2\%}$ (ksi)	σ_{ult} (ksi)
1	B-3	As Received	78.3	83.3
2	B-3		78.7	83.7
3	B-4		78.3	83.3
4	B-4		78.5	83.6
5	1A-2	355 °F (179 °C) for 2 Hours	66.4	75.2
6	1A-2		66.4	75.3
7	1B-2	355 °F (179 °C) for 1 Hour	68.3	76.8
8	1B-2		68.1	77.0
9	2A-2	320 °F (160 °C) for 2 Hours	71.6	80.1
10	2A-2		71.6	79.7
11	2B-2	320 °F (160 °C) for 1 Hour	72.5	79.4
12	2B-2		72.7	79.7
13	3A-2	285 °F (141 °C) for 2 Hours	75.1	81.7
14	3A-2		74.8	81.7
15	3B-2	285 °F (141 °C) for 1 Hour	75.2	81.3
16	3B-2		74.7	81.6
17	4A-2	250 °F (121 °C) for 2 Hours	77.6	82.6
18	4A-2		77.8	82.4
19	4B-2	250 °F (121 °C) for 1 Hour	76.7	82.5
20	4B-2		76.9	82.7

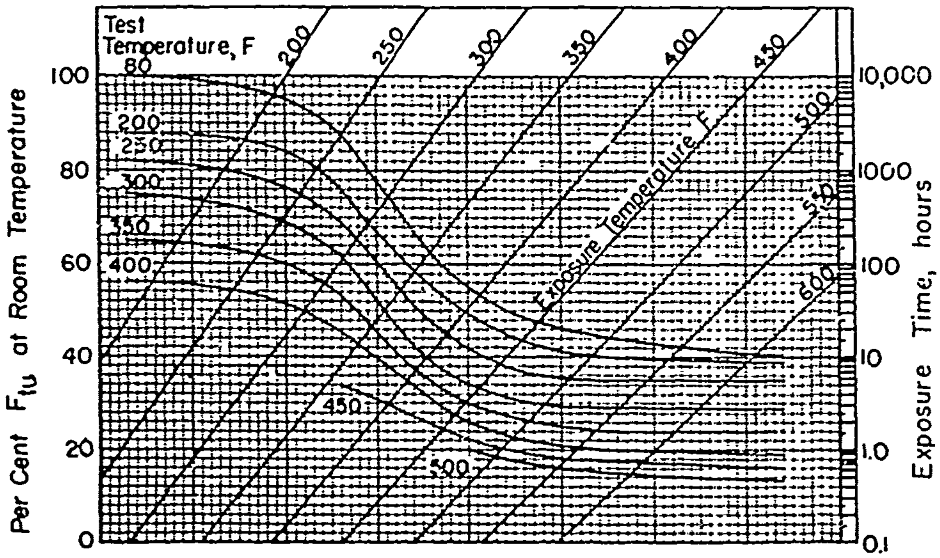


Figure C-2. Effect of Temperature on the Ultimate Tensile Strength (F_{tu}) of 7075-T6, T651, T6510, and T6511 Aluminum Alloy (All Products)*

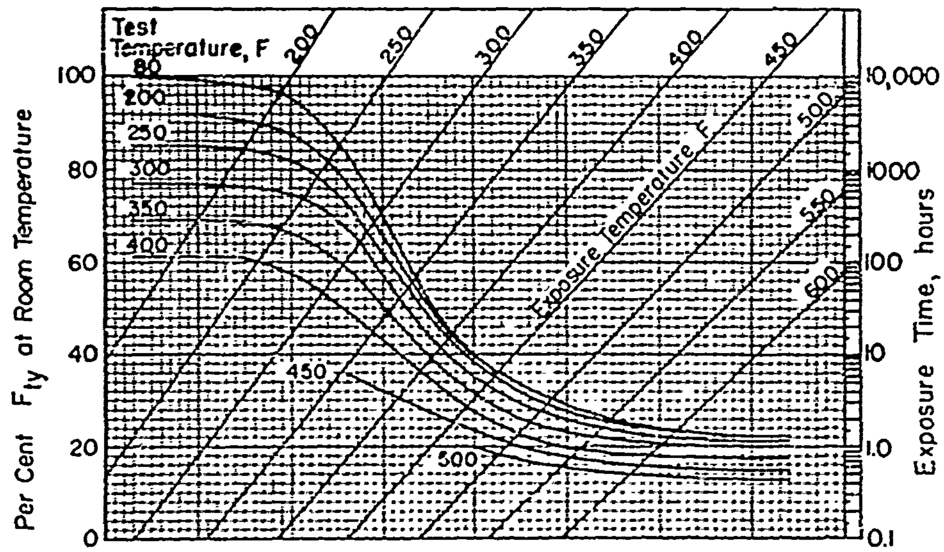


Figure C-3. Effect of Temperature on the Tensile Yield Strength (F_{ty}) of 7075-T6, T651, T6510 and T6511 Aluminum Alloy (All Products)*

*Figures 3.7.3.1.1 (a) and (b) MIL-HDBK-5C, Page 3-260

TABLE C-2

HARDNESS TESTS

TEST NUMBER	ORIGINAL SPECIMEN NUMBER	HARDNESS (ROCKWELL "B")	AVERAGE
1	B-3	91 91	91
2	B-3	89 90	89.5
3	B-4	90 91	90.5
4	B-4	88 90	89
5	1A-2	82 85	83.5
6	1A-2	82 85	83.5
7	1B-2	83 85	84
8	1B-2	82 85	83.5
9	2A-2	84 87	85.5
10	2A-2	87 87	87
11	2B-2	86 87	86.5
12	2B-2	83 87 88	86
13	3A-2	84 88 90	87.3
14	3A-2	88 90	89
15	3B-2	88 89	88.5
16	3B-2	87 90	88.5
17	4A	89 90	89.5
18	4A	89 91	90
19	4B	87 89	88
20	4B	89 89	89

APPENDIX D

CONSTANT AMPLITUDE DATA

Data from three constant amplitude tests were evaluated. The maximum applied stress was 9.9 Ksi (68 MPa) and the stress ratio was 0.5. Specimens were center-cracked panels measuring 0.25 inches (6.35 mm) thick, 3.95 inches (100 mm) wide, and 16 inches (406 mm) long. An initial notch 0.2 inches (5 mm) in length was introduced, but specimens were not precracked.

Crack growth lives were calculated in the same manner as for the flight-by-flight tests. Life was defined to be the number of cycles required for a 0.3 inch (7.62 mm) crack to grow to failure. Fatigue and crack growth lives for the three specimens are shown in Table D-1.

Raw 2a vs N data for the three constant amplitude test specimens follows Table D-1.

TABLE D-1

SPECIMEN LIVES*

SPECIMEN	CYCLES TO 0.3 INCHES (INTERPOLATED)	CYCLES TO FAILURE	CRACK GROWTH LIFE
CCP 3	209,850	463,100	252,250
CCP 4	222,375	479,700	257,325
CCP 355	176,000	457,700	281,700

* CONSTANT AMPLITUDE LOADING
MAXIMUM STRESS: 9.9 Ksi
STRESS RATIO: 0.5

CCP 3 (BASELINE)

N (CYCLES)	2a (INCHES)
0	0.209
54000	0.213
61000	0.216
68000	0.223
78000	0.228
98000	0.24
118000	0.252
138000	0.256
158000	0.266
178000	0.284
198000	0.292
218000	0.306
238000	0.322
258000	0.342
278000	0.367
298000	0.398
318000	0.442
338000	0.514
358000	0.622
368000	0.672
378000	0.742
388000	0.82
398000	0.898
408000	0.988
418000	1.098
423000	1.156
428000	1.214
433000	1.292
438000	1.368
443000	1.51
445000	1.572
447000	1.638
449000	1.712
451000	1.786
453000	1.876
455000	1.98
458000	2.17
460000	2.354
461000	2.466
462000	2.638
463000	3.057

CCP 4 (BASELINE)

N (CYCLES)	2a (INCHES)
0	0.21
47000	0.212
54000	0.213
61000	0.215
68000	0.221
78000	0.225
98000	0.229
118000	0.241
138000	0.254
158000	0.262
178000	0.274
198000	0.282
218000	0.297
238000	0.313
258000	0.329
278000	0.349
298000	0.374
318000	0.396
338000	0.454
358000	0.508
378000	0.62
388000	0.676
398000	0.742
408000	0.806
418000	0.888
428000	0.972
438000	1.076
443000	1.166
448000	1.276
450000	1.312
452000	1.372
454000	1.414
456000	1.44
458000	1.516
460000	1.578
462000	1.644
464000	1.708
466000	1.788
468000	1.868
470000	1.966
471000	2.016
472000	2.08
473000	2.137
474000	2.201
475000	2.274
476000	2.358
477000	2.47
478000	2.603
478500	2.696
479000	2.821
479500	3.048
479700	3.291

CCP 355 (355 ^{o}F, 179 ^{o}C)

N (CYCLES)	2a (INCHES)	N (CYCLES)	2a (INCHES)
0	0.209	430000	1.454
44000	0.226	434000	1.528
54000	0.23	438000	1.612
64000	0.233	442000	1.702
74000	0.239	444000	1.706
84000	0.248	446000	1.806
92000	0.253	448000	1.87
102000	0.254	450000	1.937
112000	0.262	452000	2.009
122000	0.265	454000	2.096
132000	0.271	455000	2.145
142000	0.273	456000	2.506
152000	0.28	456600	2.579
162000	0.288	457000	2.64
172000	0.297	457500	2.76
182000	0.302	457700	2.806
192000	0.311		
202000	0.32		
212000	0.327		
222000	0.342		
232000	0.352		
238000	0.356		
248000	0.37		
258000	0.394		
268000	0.414		
278000	0.434		
288000	0.452		
298000	0.482		
308000	0.512		
318000	0.554		
328000	0.594		
338000	0.638		
348000	0.694		
354000	0.724		
360000	0.766		
366000	0.796		
372000	0.838		
378000	0.88		
384000	0.932		
390000	0.984		
396000	1.036		
402000	1.092		
406000	1.134		
410000	1.176		
414000	1.222		
418000	1.274		
422000	1.33		
426000	1.39		

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